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Computer Code for Single-Point Thermodynamic Analysis of Hydrogen/Oxygen Expander-Cycle Rocket Engines

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Summary

A computer code for a single-point thermodynamic analysis of hydrogen/oxygen expander-cycle rocket engines is described in this report. Full, split, and dual expander cycles are included. Heat regeneration between the turbine exhaust and the pump exhaust is allowed. The combustion process is modeled as one of chemical equilibrium in either an infinite-area or a finite-area combustor. Gas composition in the nozzle during expansion may be either equilibrium or frozen.

This report, which serves as a users guide for the computer code, describes the system, the analysis methodology, and the program input and output and contains sample calculations to show the effects of key variables such as nozzle area ratio and oxidizer-to-fuel mass ratio. For this analysis, ideal specific impulse is a function of only nozzle area ratio and oxidizer-to-fuel mass ratio and is independent of expander-cycle type, chamber pressure, regenerator effectiveness, and turbomachine efficiencies. These latter parameters primarily affect the pump exit pressure required to achieve the specified chamber pressure.

A single-point analysis is useful for determining system performance from known component performance and for performing sensitivity analyses over narrow ranges of the cycle variables. In analyzing a given engine over a wide range of cycle variables, care must be taken to use appropriate variations in component performance parameters.

Introduction

Expander-cycle engines are the primary propulsion candidates for orbital transfer missions. The advantages of an expander-cycle engine are good specific impulse, engine simplicity, and relatively low weight (ref. 1). For a topping cycle, such as the expander cycle, however, the fuel-pump discharge pressure is much higher than that for a bleed cycle, such as the gas generator cycle, and this tends to limit the achievable combustion chamber pressure.

In support of space propulsion studies being conducted at the NASA Lewis Research Center, a computer code for the single-point thermodynamic analysis of hydrogen/oxygen expander-cycle rocket engines was developed. This code is applicable to full, split, and dual expander cycles, which can include a regenerator. Vacuum specific impulse, cycle temperatures, and cycle pressures are computed. Real fluid properties and methodologies are used for all computations except combustion and expansion. Combustion and nozzle expansion are modeled with one-dimensional, ideal-gas calculations. The combustion process is one of chemical equilibrium in either an infinite-area or a finite-area chamber. Expansion in the nozzle may be either an equilibrium or a frozen chemical process.

This report presents the cycle analysis methodology and also serves as a users guide for the computer code. The expander cycles are described and the cycle analysis equations are presented. Program input and output are described and sample cases are included. The sample calculations are intended to show the effects of some of the key variables on the cycle performance.

System Description

The computer code presented herein can be used to analyze the single-point performance of full, split, and dual expander rocket engine cycles. Each of these cycles can incorporate a regenerator on the fuel side to transfer heat from the turbine exit stream to the pump exit stream. The three cycles are described in this section. The format for the system sketches is that used by Pratt & Whitney in reference 2.

The full expander cycle with a regenerator is shown in figure 1. Liquid hydrogen is pumped to high pressure and then flows through the regenerator, where heat is transferred to it from the turbine exit stream. The hydrogen is further heated as it regeneratively cools the nozzle and the combustion chamber. Most of the heated hydrogen then expands in series through the oxygen-pump and hydrogen-pump drive turbines. Alternatively, the order of the flow through the turbines can be reversed or the flow can be in parallel. Some of the heated hydrogen from the combustion chamber cooling jacket is bypassed around the turbines (and the regenerator) to provide a means for controlling turbine power. After transferring heat to the pump exit stream in the regenerator, the turbine exit flow mixes with the bypass flow, passes through the fuel shutoff valve, and enters the combustion chamber. On the oxidizer side the liquid oxygen is pumped to the pressure required to match the hydrogen pressure at the combustor. It then flows through the oxidizer control valve, which controls the oxidizer-to-fuel (o/f) ratio, and enters the combustion chamber, where combustion occurs. The combustion gases then expand through the nozzle to high velocity to provide the thrust.

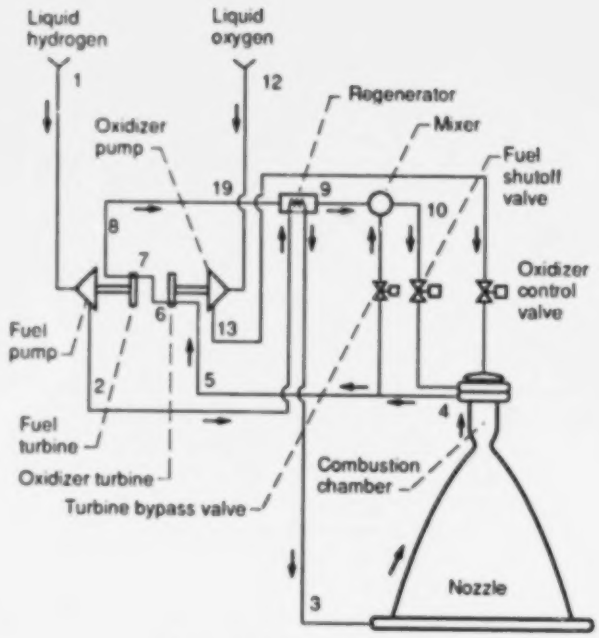

Figure 1.—Full expander cycle.

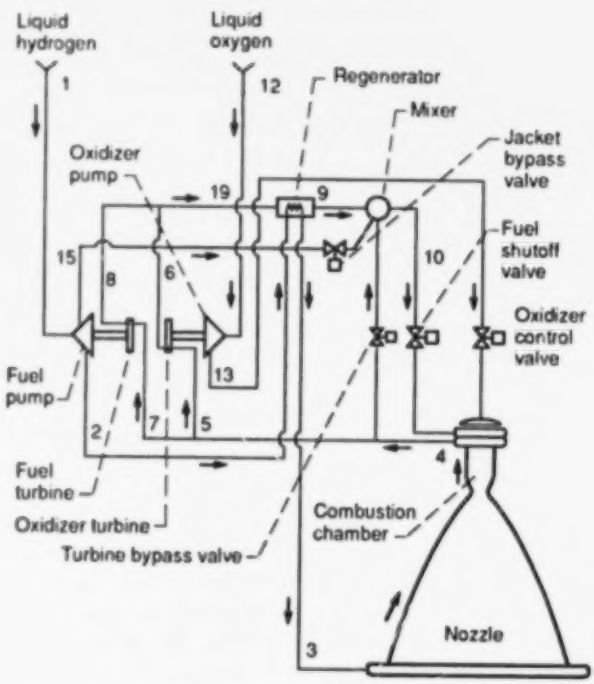

Figure 2.—Split expander cycle.

A split expander cycle is shown in figure 2. This cycle derives its name from the fact that the hydrogen flow is split in the fuel-pump system. Only part of the hydrogen is pumped to maximum pressure for use as the chamber/nozzle regenerative coolant and the turbine drive fluid. This figure shows

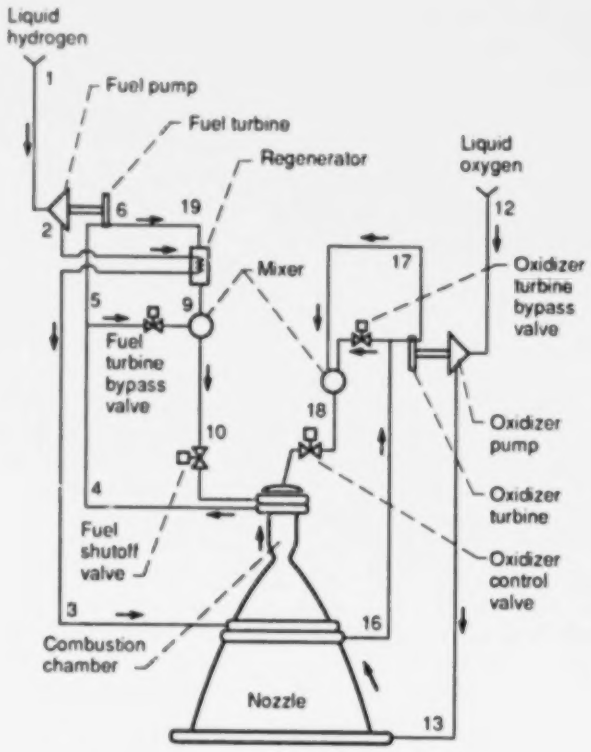

Figure 3.—Dual expander cycle.

parallel flow through the turbines, which is one of the available modeling options. The remaining hydrogen flows through the jacket bypass valve, which controls the flow split, and mixes with the turbine exhaust and bypass streams before entering the combustion chamber. A regenerator can be incorporated into this cycle in a similar manner as for the full expander cycle (fig. 1).

A dual expander cycle is shown in figure 3. In this cycle each of the fluids is heated by regenerative cooling of the nozzle or the combustion chamber, or both, and used in a separate turbine to drive its own pump. Thus, the fluids can be kept separated before combustion without relying on the turbopump seals. A bypass valve controls the flow to each turbine. If desired, a regenerator can be incorporated on the fuel side of the cycle.

The method of analysis is documented later in this report. Symbols used for the analysis are defined in appendix A, and the analysis method is presented in appendix B.

Program Input and Output

Input

The input data consist of namelist INPUT, which is required for the cycle calculations, and THERMO data, REACTANTS data, and namelists INPT2 and RKTINP, which are used for

the combustion and expansion calculations performed by the Gordon and McBride CEC code (refs. 3 and 4). The CEC input data are described in detail in references 3 and 4 and are described here only to the extent necessary to run this program without consulting the CEC references. Input for a sample case is presented in figure 4. The output corresponding to this sample input is presented and described in the next section.

Namelist INPUT is read from unit 2. The INPUT variables along with descriptions, units, and any special instructions are presented in the following list:

P1	fuel-pump inlet pressure (saturation temperature is assumed), psi
P12	oxidizer-pump inlet pressure (saturation temperature is assumed), psi
PC	combustion chamber injection pressure, psi
TBP	fuel-turbine bypass fraction
OBP	oxidizer-turbine bypass fraction (dual expander only)
JBP	cooling-jacket bypass fraction (split expander only)
TC	thrust correction factor, ratio of actual thrust to ideal thrust
OF	oxidizer-to-fuel flow ratio
ETAFP	fuel-pump efficiency
ETAFT	fuel-pump drive turbine efficiency
ETALP	oxidizer-pump efficiency
ETALT	oxidizer-pump drive turbine efficiency
ETABP	low-pressure, fuel-pump efficiency (split expander only)
EPS	regenerator effectiveness (input only for IREG = 1)

```
THERMO                                                                                                  Formats
   298.15   1000.000  5000.000                                                                          3A4
H                 J 3/77H  1.   0.   0.   0.G   300.000  5000.000      1.00790R1                       3F10.3
 0.25000000E 01     0.00000000     0.00000000      0.00000000      0.00000000     2                    3A4,5(2A3,4(A2,F3.0,A,2F7.0,F15.5),I2)
 0.25474390E 05-0.45989841E 00 0.25000000E 01      0.00000000      0.00000000     3 }                  5(E15.8,I5)
    0.00000000      0.00000000 0.25474390E 05-0.45989841E 00      0.00000000     4 }
HO2               J 9/78H  1.0  2.   0.   0.G   300.000  5000.000     33.00670 1
 0.40173060E 01 0.22175883E-02-0.57710171E-06 0.71372882E-10-0.36458591E-14     2
-0.11412445E 04 0.37846051E 01 0.35964102E 01 0.52500748E-03 0.75118344E-05     3
-0.95674952E-08 0.36597628E-11-0.89333502E 03 0.66372671E 01     0.00000000     4
H2                J 3/77H  2.   0.   0.   0.G   300.000  5000.000      2.01580 1
 0.30558123E 01 0.59740400E-03-0.16747471E-08-0.21247544E-10 0.25195487E-14     2
-0.86168476E 03-0.17207073E 01 0.29432327E 01 0.34815509E-02-0.77713819E-05     3
 0.74997496E-08-0.25203379E-11-0.97695413E 03-0.18186137E 01     0.00000000     4
H2O               J 3/79H  2.0  1.   0.   0.G   300.000  5000.000     18.01520 1
 0.26340654E 01 0.31121899E-02-0.90278449E-06 0.12673054E-09-0.69164732E-14     2
-0.29876258E 05 0.70823873E 01 0.41675564E 01-0.18106868E-02 0.59450878E-05     3
-0.48670871E-08 0.15284144E-11-0.30289546E 05-0.73087997E 00     0.00000000     4
H2O2              L 3/85H  2.0  2.   0.   0.G   300.000  5000.000     34.01460 1
 0.47928858E 01 0.36300865E-02-0.11136435E-05 0.14868513E-09-0.68958511E-14     2
-0.18132195E 05-0.51306415E 00 0.34546633E 01 0.55575930E-02 0.92103738E-06     3
-0.46279780E-08 0.21458200E-11-0.17672328E 05 0.68402452E 01-0.16394994E 05     4
O                 J 3/77O  1.   0.   0.   0.G   300.000  5000.000     15.99940R1
 0.25342961E 01-0.12478170E-04-0.12562724E-07 0.69029862E-11-0.63797095E-15     2
 0.29231108E 05 0.49628591E 01 0.30309401E 01-0.22525853E-02 0.39824540E-05     3
-0.32604921E-08 0.10152035E-11 0.29136526E 05 0.26099342E 01     0.00000000     4
OH                J 6/77O  1.H  1.   0.   0.G   300.000  5000.000     17.00730R1
 0.28897814E 01 0.10005879E-02-0.22048807E-06 0.20191288E-10-0.39409831E-15     2
 0.38857042E 04 0.55566427E 01 0.38737300E 01-0.13393772E-02 0.16348351E-05     3
-0.52133639E-09 0.41826974E-13 0.35802348E 04 0.34202406E 00     0.00000000     4
O2                J 3/77O  2.   0.   0.   0.G   300.000  5000.000     31.99880 1
 0.36122139E 01 0.74853166E-03-0.19820647E-06 0.33749008E-10-0.23907374E-14     2
-0.11978151E 04 0.36703307E 01 0.37837135E 01-0.30233634E-02 0.99492751E-05     3
-0.98189101E-08 0.33031825E-11-0.10638107E 04 0.36416345E 01     0.00000000     4
O3                J 6/61O  3.0  0.0  0.0  0.G   300.000  5000.000     47.99820R1
 0.54665239E 01 0.17326031E-02-0.72204889E-06 0.13721660E-09-0.96233828E-14     2
 0.15214096E 05-0.34712616E 01 0.24660617E 01 0.91703209E-02-0.49698480E-05     3
-0.20634230E-08 0.20015595E-11 0.16059556E 05 0.12172130E 02     0.00000000     4
H2O(L)            J 3/79H  2.0  1.   0.   0.L   273.150   500.000     18.01520 1
 0.28630800E 02-0.20260986E 00 0.78529479E-03-0.13653020E-05 0.91326966E-09     2
-0.38579539E 05-0.11895046E 03 0.28630800E 02-0.20260986E 00 0.78529479E-03     3
-0.13653020E-05 0.91326966E-09-0.38579539E 05-0.11895046E 03     0.00000000     4
END
REACTANTS                                                                                               3A4
H 2.                                          100.00  - 100.00 L 270.35 F                               A2,F9.3,F7.2,19(A
O 2.                                          100.00  -2000.00 L 129.86 O

NAMELISTS
 &INPUT
  P1=18.0,P12=16.0,PC=1500.,TBP=0.05,TC=1.00,DPCREG=0.02,DPHPEG=0.04,
  ETAFP=0.65,ETAFT=0.85,ETALP=0.70,ETALT=0.85,EPS=0.4,
  DPFI=0.15,DPLI=0.35,DTRISE=450.0,DPLOSS=0.15,
  OF=6.5,CASE=1,IHO=1,IREG=2,IDBG=0,
 &END
 &INPT2 P=1500.,MIX=6.5 &END

 &RKTINP EQL=.TRUE.,FROZ=.TRUE.,FAC=.TRUE.,ACAT=2.5,SUPAR=1000.,&END
STOP
```

Figure 4 – Sample input

DPCREG regenerator cold-side fractional pressure loss (input only for IREG = 1)

DPHREG regenerator hot-side fractional pressure loss (input only for IREG = 1)

DPFI fuel-stream (line valve injector) fractional pressure loss

DPLI oxidizer-stream (line valve injector) fractional pressure loss

DPBI bypass-stream (line valve injector) fractional pressure loss (split expander only)

DPLOSS nozzle chamber cooling jacket fractional pressure loss (full and split expanders only)

DTRISE nozzle chamber coolant temperature rise, °R (full and split expanders only)

DPCHM chamber cooling jacket (fuel) fractional pressure loss (dual expander only)

DPNOZ nozzle cooling jacket (oxidizer) fractional pressure loss (dual expander only)

DTCHM chamber coolant (fuel) temperature rise, °R (dual expander only)

DTNOZ nozzle coolant (oxidizer) temperature rise, °R (dual expander only)

CASE cycle-type indicator

1—full expander cycle

2—split expander cycle

3—dual expander cycle

IREG regenerator indicator

1—no regenerator

2—fuel-side regenerator

IHO turbine arrangement indicator (full and split expanders only)

1—series with fuel turbine followed by oxidizer turbine

2—series with oxidizer turbine followed by fuel turbine

3—parallel

IDBG debug output indicator

0—no additional output

1—values of pressure p, absolute temperature t, and enthalpy h for each station printed during calculations for both unconverged and converged passes

The CEC input data are read from unit 5 and consist of THERMO data, REACTANTS data, and namelists INPT2 and RKTINP. The THERMO data are the least-squares, curve-fit coefficients of the thermodynamic data for the species H, HO_2, H_2, H_2O (vapor and liquid), H_2O_2, O, OH, O_2, and O_3. These are the species considered for the equilibrium calculations for the H_2/O_2 system. These THERMO data are used exactly as presented in figure 4. Formats are included in the table. Each time the THERMO data are read from unit 5, they are written in binary format to unit 4 and subsequently read from unit 4 for use in the program. Therefore, once the binary file exists, the THERMO data need not be included in the input data read from unit 5.

The REACTANTS data identify each of the reactant species. These data are used exactly as presented in figure 4. Formats are included in the figure.

The only variables to be included in namelist INPT2 are

P must be set equal to value of PC in namelist INPUT

MIX must be set equal to value of OF in namelist INPUT

The variables that need be included in namelist RKTINP are

EQL logical variable set to .TRUE. for equilibrium chemical composition in the rocket nozzle

FROZ logical variable set to .TRUE. for frozen chemical composition in the rocket nozzle (defaults to .FALSE. if FAC = .TRUE.)

FAC logical variable specifying combustion process
.TRUE.—finite-area combustion, which includes total-pressure loss due to heat addition
.FALSE.—infinite-area combustion

ACAT ratio of combustor area to throat area (input only if FAC = .TRUE.)

SUPAR rocket-nozzle supersonic area ratio (exit area to throat area)

Output

Program output consists of normal output, diagnostic output, and error messages.

Normal output.—Normal output, which is written to unit 7, consists of a listing of input variable values, gas composition after both combustion and expansion, state conditions at all cycle points, and rocket performance. Figure 5 presents the normal output that corresponds to the sample input shown in figure 4.

System configuration and input values are presented descriptively in the output (fig. 5). This particular case is for a full expander cycle with regenerator. The key parameters are a combustion chamber injection pressure of 1500 psia, an oxidizer-to-fuel ratio of 6.5, and a nozzle area ratio of 1000. All the component efficiencies and pressure losses are shown.

This is specified to be a case with finite-area combustion and equilibrium expansion. Equilibrium gas compositions in terms of species mole fractions are shown for the combustor exit and the nozzle exit. The gas temperature at the nozzle exit is sufficiently low that the only species there are unburned hydrogen and steam. If the expansion were specified to be frozen, the nozzle exit composition would be the same as the combustion chamber composition.

```
                    ROCKET  ENGINE  CYCLE  CODE
              FULL EXPANDER CYCLE WITH REGENERATOR

FUEL PRESSURE =  18.0 PSIA
OXYGEN PRESSURE =  16.0 PSIA
INJECTION PRESSURE = 1500.0 PSIA
THRUST CORRECTION FACTOR = 1.00
FUEL TURBINE BYPASS RATIO = 0.05
O/F = 6.5

COMPONENT EFFICIENCIES:
H2 PUMP   H2 TURBINE   O2 PUMP   O2 TURBINE   H2 LP PUMP   REGENERATOR
 0.650      0.850       0.700      0.850        1.000        0.40

NOZZLE JACKET DELTA T = 450.0 R.
NOZZLE JACKET (DELTA P)/P = 0.15
REGENERATOR (DELTA P)/P (HOT AND COLD SIDES) = 0.04 AND 0.02
FUEL INJECTOR (DELTA P)/P = 0.15
LOX INJECTOR (DELTA P)/P = 0.35

OPTIONS:
CASE = 1     IREG = 2     IHO = 1


EQUILIBRIUM MOLE FRACTIONS IN CHAMBER AND AT EXIT:
H                              0.03274      0.00000
HO2                            0.00010      0.00000
H2                             0.20194      0.18102
H2O                            0.69675      0.81898
H2O2                           0.00002      0.00000
O                              0.00522      0.00000
OH                             0.05619      0.00000
O2                             0.00704      0.00000

                                                    CYCLE VALUES
                            PRESSURE     TEMPERATURE      FLOW      HORSEPOWER
                              PSIA       DEGREES R       LB/SEC
  ********** FUEL **********
FUEL PUMP INLET :              18.00         37.75        1.000
FUEL HP PUMP  EXIT :         3322.14         79.62        1.000       278.1
REGENERATOR COLD EXIT:       3255.70        327.53        1.000
COOLING JACKET EXIT :        2767.34        757.58        1.000
FUEL TURBINE EXIT :          2018.09        703.24        0.950       278.1
LOX TURBINE EXIT :           1838.29        687.73        0.950        77.3
REGENERATOR HOT EXIT :       1764.76        436.78        0.950
  TURBINE BYPASS :           2767.34        757.58        0.050
MIXER EXIT :                 1764.76        452.55        1.000
FUEL INJECTOR EXIT :         1500.05        453.17        1.000
  ******** OXIDIZER ********
LOX PUMP INLET :               16.00        163.80        6.500
LOX PUMP EXIT :              2307.69        175.54        6.500        77.3
LOX INJECTOR EXIT :          1500.00        178.72        6.500
  ***** CHAMBER/NOZZLE *****
CHAMBER STAGNATION, EQL :    1452.30       6479.23
NOZZLE EXIT STATIC, EQL :       0.05       1191.71
EQUILIBRIUM PERFORMANCE PARAMETERS:
      NOZZLE AREA RATIO = 1000.
      CHARACTERISTIC VELOCITY = 7476. FT/SEC.
      NOZZLE EXIT MACH NUMBER =  6.912 .
      CHAMBER MACH NUMBER =  0.246 .
      VACUUM SPECIFIC IMPULSE =  493.4 LBF-SEC/LBM
```

Figure 5 – Sample output

Temperatures, pressures, and flow rates are shown for all cycle calculation stations. Flow rates are based on 1 lb/sec of fuel, since size effects are not considered in this single-point thermodynamic analysis. Also shown are turbomachinery power requirements. For this case, a fuel-pump exit pressure of 3322 psia is required to provide the specified combustion chamber injection pressure of 1500 psia. The finite-area combustion, which occurred at a chamber Mach number of 0.246, resulted in a reduction in chamber stagnation pressure from 1500 to 1452 psia. Rocket performance is presented in

```
P11 AND P COMBUSTION ARE :   1632.03418        1500.00000
P11 AND P COMBUSTION ARE :   1577.80420        1500.00000
P11 AND P COMBUSTION ARE :   1521.67896        1500.00000
P11 AND P COMBUSTION ARE :   1463.55029        1500.00000
P11 AND P COMBUSTION ARE :   1500.04761        1500.00000
```

Figure 6.—Chamber pressure convergence.

```
 P11 AND P COMBUSTION ARE :   1632.03418        1500.00000
 FUEL PUMP PRESSURE RATIO IS   208.333328
FUEL PUMP EXIT :          3569.99976      82.6108398      103.625839
NOZZLE JACKET ENTR :       3498.59961      307.661865      953.772949
NOZZLE JACKET EXIT :       2973.80957      757.661865      2624.23120
FUEL TURBINE INLET :       2973.80957      757.661865      2624.23120
POW1 AND DHTURB1 ARE       210.422180      223.490814
POW1 AND DHTURB1 ARE       210.422180      209.631226
POW1 AND DHTURB1 ARE       210.422180      210.469269
POW1 AND DHTURB1 ARE       210.422180      210.423935
FUEL TURBINE EXIT :        2122.30225      699.821533      2402.73218
LOX TURBINE INLET :        2122.30225      699.821533      2402.73218
POW2 AND DHTURB2 ARE  55.0481567          56.9240112
POW2 AND DHTURB2 ARE  55.0481567          54.9870758
POW2 AND DHTURB2 ARE  55.0481567          55.0491791
LOX TURBINE EXIT :        1932.98535      684.376953      2344.78564
NOZZLE JACKET ENTR :       3498.59961      308.018555      955.266357
NOZZLE JACKET EXIT :       2973.80957      758.018555      2625.48853
FUEL TURBINE INLET :       2973.80957      758.018555      2625.48853
POW1 AND DHTURB1 ARE       210.422180      223.496719
POW1 AND DHTURB1 ARE       210.422180      209.625900
POW1 AND DHTURB1 ARE       210.422180      210.471451
POW1 AND DHTURB1 ARE       210.422180      210.418610
FUEL TURBINE EXIT :        2122.64673      700.178467      2403.99512
LOX TURBINE INLET :        2122.64673      700.178467      2403.99512
POW2 AND DHTURB2 ARE  55.0481567          56.9275665
POW2 AND DHTURB2 ARE  55.0481567          54.9809570
POW2 AND DHTURB2 ARE  55.0481567          55.0568542
POW2 AND DHTURB2 ARE  55.0481567          55.0477905
LOX TURBINE EXIT :        1933.39917      684.733398      2346.05005
NOZZLE JACKET ENTR :       3498.59961      308.133545      955.746826
NOZZLE JACKET EXIT :       2973.80957      758.133545      2625.89307
FUEL TURBINE INLET :       2973.80957      758.133545      2625.89307
POW1 AND DHTURB1 ARE       210.422180      223.490814
POW1 AND DHTURB1 ARE       210.422180      209.626694
POW1 AND DHTURB1 ARE       210.422180      210.470078
POW1 AND DHTURB1 ARE       210.422180      210.424728
FUEL TURBINE EXIT :        2122.72705      700.291016      2404.39221
LOX TURBINE INLET :        2122.72705      700.291016      2404.39221
POW2 AND DHTURB2 ARE  55.0481567          56.9182892
POW2 AND DHTURB2 ARE  55.0481567          54.9898224
POW2 AND DHTURB2 ARE  55.0481567          55.0497742
LOX TURBINE EXIT :        1933.47485      684.844971      2346.44604
NOZZLE JACKET ENTR :       3498.59961      308.169434      955.397217
NOZZLE JACKET EXIT :       2973.80957      758.169434      2626.02002
FUEL TURBINE INLET :       2973.80957      758.169434      2626.02002
POW1 AND DHTURB1 ARE       210.422180      223.495529
POW1 AND DHTURB1 ARE       210.422180      209.635361
POW1 AND DHTURB1 ARE       210.422180      210.464752
POW1 AND DHTURB1 ARE       210.422180      210.423935
FUEL TURBINE EXIT :        2122.79712      700.326416      2404.52100
LOX TURBINE INLET :        2122.79712      700.326416      2404.52100
POW2 AND DHTURB2 ARE  55.0481567          56.9277496
POW2 AND DHTURB2 ARE  55.0481567          54.9912109
POW2 AND DHTURB2 ARE  55.0481567          55.0468140
LOX TURBINE EXIT :        1933.58374      684.881836      2346.57690
 P11 AND P COMBUSTION ARE :   1577.80420        1500.00000
 FUEL PUMP PRESSURE RATIO IS   198.333328
```

Turbine iterations to convergence

Regenerator iteration loop

Fuel-pump exit pressure iteration loop

Figure 7.—Detailed diagnostic output.

terms of characteristic velocity and vacuum specific impulse, which are 7476 ft/sec and 493 sec, respectively, for this case. Also shown is the nozzle exit Mach number, which is 6.9.

Diagnostic output.—Diagnostic output is written to unit 6 in two forms. A tracking of the convergence of combustion chamber injection pressure to the required input value is always printed as shown in figure 6. If the input variable IDBG = 1, a longer diagnostic output, as shown in figure 7, tracking the entire computation is printed. Pressures, temperatures, and enthalpies are shown for the various calculation stations, and required-versus-calculated powers are shown for the turbines. Figure 7 shows one fuel-pump exit pressure iteration loop. The regenerator iteration loop is indicated as are the turbine iterations to convergence.

Error messages.—The program contains three output messages that indicate abnormal operation due to either the nonexistence of a solution satisfying the specified input requirements or the inability of a convergence algorithm to find a

solution. These messages are presented in this section, and their causes are discussed. In general, when one of the following messages appears, the program input should be checked for errors.

(1) TURBINE N NOT CONVERGED'—The value of N is either 1 or 2 depending on which turbine in the sequence is not converged. This message is caused either by insufficient energy in the turbine drive fluid or by a failure in the convergence scheme. Corrective action includes reducing input chamber pressure or increasing turbine drive fluid energy, or both.

(2) NO SOLUTION FOR CHAMBER PRESSURE. MAX PC = XXXXX.—This message is caused by the maximum achievable chamber pressure (shown in error message) being less than the chamber pressure specified by program input. If the input design parameters are correct, the input chamber pressure must be reduced.

(3) CHAMBER PRESSURE SOLUTION NOT FOUND IN 100 ITERS—This message is caused by the program making 100 iterations without either finding a pump exit pressure that yields the required chamber pressure or determining that none exists. This situation has not occurred and there is no obvious reason for it to occur, but the possibility does exist.

Sample Calculations

Sample calculations were made to illustrate the effects of some key system parameters on cycle performance and operating conditions. The effects of nozzle area ratio and oxidizer-to-fuel mass ratio on ideal vacuum specific impulse were explored along with the effects of chamber pressure and regenerator effectiveness on pump exit pressure.

The effect of nozzle area ratio (i.e., the ratio of exit area to throat area) on ideal vacuum specific impulse is presented in figure 8. Ideal specific impulse increased very rapidly with increasing area ratio up to an area ratio of about 200, after which the increase tapered off. Going from an area ratio of 1000 to one of 2000 resulted in a 1-percent increase in specific impulse.

The effect of oxidizer-to-fuel mass ratio on ideal vacuum specific impulse is presented in figure 9. Maximum specific impulse occurred at an *o/f* ratio of about 6.5. Although combustion temperature was maximum at the stoichiometric *o/f* ratio of 8, the increase in reaction-product molecular weight with increasing *o/f* ratio caused the exit velocity, and therefore the specific impulse, to reach a maximum at an *o/f* ratio somewhat less than stoichiometric.

Figures 8 and 9 were discussed without reference to a specific cycle (full, split, or dual) because those results were independent of cycle type. For this analysis, ideal specific impulse is a function of only nozzle area ratio and o f ratio and is independent of cycle type, chamber pressure, regenerator effectiveness, and turbomachine efficiencies. These latter

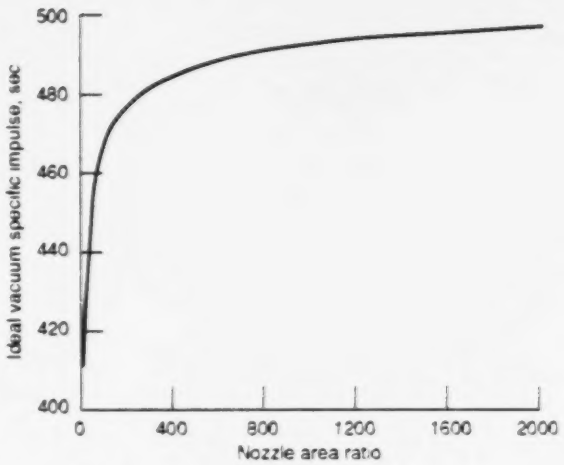

Figure 8.—Effect of nozzle area ratio on ideal vacuum specific impulse. Chamber pressure, 1500 psia; oxidizer-to-fuel ratio, 6.5.

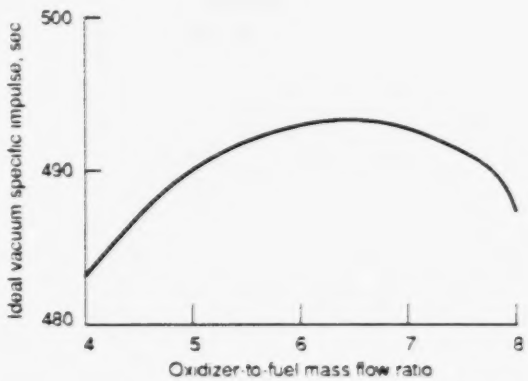

Figure 9.—Effect of oxidizer-to-fuel mass flow ratio on ideal vacuum specific impulse. Chamber pressure, 1500 psia; nozzle area ratio, 1000.

parameters primarily affect pump exit pressure, as illustrated in the next figure.

The effects of chamber pressure and regenerator effectiveness on fuel-pump exit pressure for the full expander cycle are presented in figure 10. Increasing the chamber pressure obviously required higher pump exit pressures. Increasing the regenerator effectiveness allowed a lower pump exit pressure for a specified chamber pressure or, conversely, a higher chamber pressure for a given pump exit pressure. For example, a chamber pressure of 1500 psia required a 3860-psia pump exit pressure with no regeneration but only a 2800-psia pump exit pressure with regeneration of 0.6 effectiveness. Turbine inlet temperature increased with regeneration, thus reducing the turbine pressure drop required to provide a given power output.

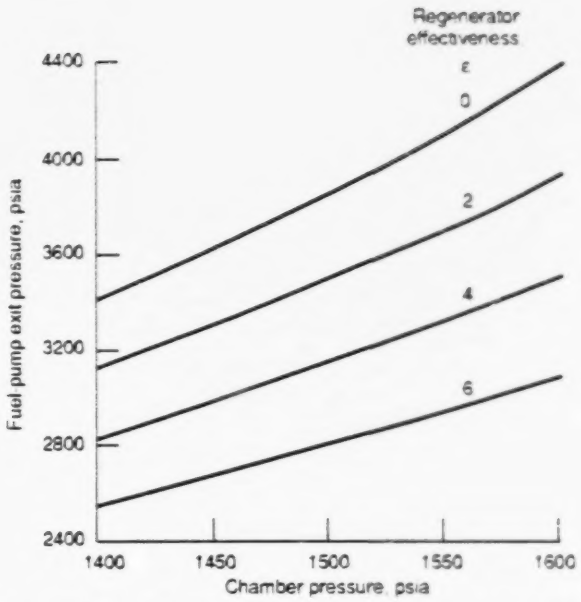

Figure 10. Effects of chamber pressure and regenerator effectiveness on fuel pump exit pressure (full expander cycle). Oxidizer-to-fuel ratio, 6.5; nozzle area ratio, 1000.

Concluding Remarks

In a single-point thermodynamic analysis all component performance parameters (i.e., turbomachine efficiencies, heat transfer effectiveness, and fractional pressure losses) are input as single values. Such an analysis, with the component parameters remaining constant, is used to obtain system performance at a single point where the component performances are known. It is also useful for performing sensitivity analyses over narrow ranges of the cycle variables where component performance would remain essentially constant.

Over a wide range of cycle variables and system size, it would be expected that component performance would vary as the component design and off-design conditions (e.g., chamber pressure, pressure ratio) varied. Therefore, for accurate analysis of an engine over a wide range of conditions, variations in the component performance parameters, as determined externally to this code, should be included as part of the program input. Future enhancements of this code could include incorporation of algorithms to correlate component performance with the cycle variables.

Summary of Results

This report presents a single-point thermodynamic code for analyzing hydrogen-oxygen expander-cycle rocket engines. The code is capable of analyzing full, split, and dual expander cycles. Heat regeneration between the turbine exhaust and the pump exhaust is allowed. The combustion process is modeled as one of chemical equilibrium in either an infinite-area or a finite-area combustor. Expansion in the nozzle can be either equilibrium or frozen.

This report, which serves as a users guide for the computer code, describes the system, the analysis methodology, and the input and output. Sample calculations are included to show the effects of some key variables. These sample calculations showed

1. That ideal specific impulse for this analysis is a function of only nozzle area ratio and oxidizer-to-fuel mass ratio and is independent of expander-cycle type, chamber pressure, regenerator effectiveness, and turbomachine efficiencies.

2. That cycle type, chamber pressure, regenerator effectiveness, and turbomachine efficiencies primarily affect the pump exit pressure required to achieve the specified chamber pressure.

This single-point analysis is useful for determining system performance from known component performance and for performing sensitivity analyses over narrow ranges of cycle conditions. For the analysis of an engine over a wide range of cycle variables, appropriate variations in component performance should be included.

Appendix A
Symbols

H	GASPLUS function for enthalpy, Btu/lb
h	enthalpy, Btu/lb
o/f	oxidizer-to-fuel mass flow ratio
P	power, Btu/sec
p	pressure, psi
S	GASPLUS function for entropy, Btu/(lb)(°R)
s	entropy, Btu/(lb)(°R)
T	GASPLUS function for temperature, °R
t	absolute temperature, °R
w	mass flow rate, lb/sec
x	mass fraction
ϵ	regenerator effectiveness
η	efficiency

Subscripts:

b	bypass
CEC	referring to CEC code
f	fuel
fcj	fuel cooling jacket
fp	fuel pump
ft	fuel turbine
ftb	fuel turbine bypass
id	ideal
inj	injection
jbp	jacket bypass
loss	loss
lp	low pressure
o	oxidizer
ocj	oxidizer cooling jacket
op	oxidizer pump
ot	oxidizer turbine
otb	oxidizer-turbine bypass
rc	regenerator cold side
rh	regenerator hot side
sat	saturated
t	turbine
0–19	station identification (see figs. 1–3)

Appendix B
Analysis Method

This analysis makes use of the GASPLUS properties program (ref. 5) available at NASA Lewis. The thermodynamic property functions obtained from GASPLUS and incorporated in the computer program are hereinafter expressed in the form

$$y = F(z_1, z_2) \tag{1}$$

where y is the value of the function F, and z_1 and z_2 are the independent variables. These are real-fluid properties and are used for all computations except those for combustion and nozzle expansion. The previously referenced ideal-gas equilibrium analysis code (CEC code) of Gordon and McBride (refs. 3 and 4) is used for the combustion and expansion analyses.

The analysis method for the full expander cycle, as shown in figure 1, is presented in full. Then, the modifications required for the split and dual expander cycles are presented. The calculation procedure for these cycles, particularly for the fuel system, is iterative, involving two or three levels of iteration depending on whether there is regeneration. For the oxidizer system, only the dual expander cycle requires such iteration. A calculation logic diagram illustrating the nature of this iterative procedure for the fuel system is shown in figure 11.

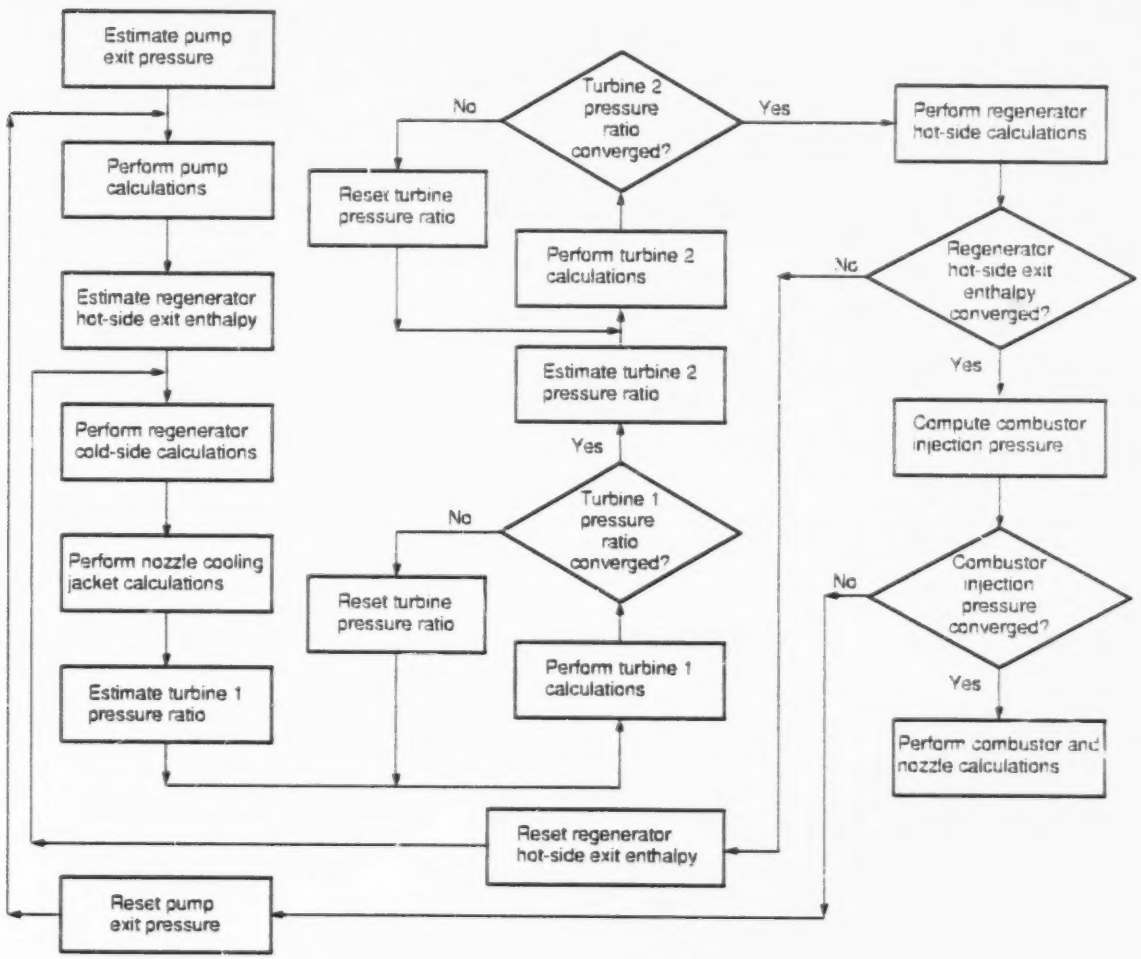

Figure 11.—Fuel system calculation logic.

Full Expander

Starting on the oxidizer side, oxygen enters the system at input pressure p_{12} and saturation temperature t_{12}, where

$$t_{12} = T_{sat}(p_{12}) \tag{2}$$

Oxidizer enthalpy, entropy, and flow rate are then given by

$$h_{12} = H(t_{12}, p_{12}) \tag{3}$$

$$s_{12} = S(t_{12}, p_{12}) \tag{4}$$

and

$$w_{12} = (o/f)w_1 \tag{5}$$

where o/f is the input oxidizer-to-fuel mass flow ratio and w_1 is the fuel mass flow rate, whose value is set to unity for this analysis.

In the oxidizer pump the oxygen is pumped to pressure p_{13}, where

$$p_{13} = \frac{p_{im}}{\left(1 - \dfrac{\Delta p_{o\,loss}}{p_{13}}\right)} \tag{6}$$

The combustion chamber injection pressure p_{im} and the oxygen-stream pressure loss ratio $\Delta p_{o\,loss}/p_{13}$, which represents the sum of the line, valve, and injector pressure losses, are both input. Pump exit conditions are then determined from

$$h_{13,ad} = H(s_{12}, p_{13}) \tag{7}$$

$$\Delta h_{op} = \frac{h_{13,ad} - h_{12}}{\eta_{op}} \tag{8}$$

$$h_{13} = h_{12} + \Delta h_{op} \tag{9}$$

$$t_{13} = T(h_{13}, p_{13}) \tag{10}$$

Assuming the pressure loss $\Delta p_{o\,loss}$ to occur at constant enthalpy, the oxygen enthalpy and temperature after injection into the combustion chamber at pressure p_{im} are

$$h_{o\,im} = h_{13} \tag{11}$$

and

$$t_{o\,im} = T(h_{o\,im}, p_{im}) \tag{12}$$

On the fuel side, hydrogen enters the system with flow rate w_1 at input pressure p_1 and saturation temperature t_1, where

$$t_1 = T_{sat}(p_1) \tag{13}$$

Fuel enthalpy and entropy are then given by

$$h_1 = H(t_1, p_1) \tag{14}$$

and

$$s_1 = S(t_1, p_1) \tag{15}$$

The hydrogen is pumped to pressure p_2 in the fuel pump. A converged value of p_2 must be determined iteratively on the basis of the calculated hydrogen pressure after injection matching the input value of combustion chamber injection pressure p_{im}. On the basis of an initially estimated value of p_2, the fuel-pump exit conditions are determined as

$$h_{2,ad} = H(s_1, p_2) \tag{16}$$

$$\Delta h_{fp} = \frac{h_{2,ad} - h_1}{\eta_{fp}} \tag{17}$$

$$h_2 = h_1 + \Delta h_{fp} \tag{18}$$

$$t_2 = T(h_2, p_2) \tag{19}$$

The thermal performance of the regenerator is specified by its effectiveness, which is an input and is defined as

$$\varepsilon = \frac{w_2(h_3 - h_2)}{w_{ha}(h_{ha} - h_2)} = \frac{h_{ha} - h_b}{h_{ha} - h_2} \tag{20}$$

where the hot-side flow rate w_{ha} is less than the cold-side flow rate w_2 as a result of the fuel-turbine bypass ratio x_{tb}, which is input.

$$w_{ha} = (1 - x_{tb})w_2 \tag{21}$$

Since the value of h_3 depends on the value of h_{ha}, which is unknown at this point, these values must be determined iteratively. With an initial estimate for h_{ha}, we get

$$h_3 = h_2 + \varepsilon\left(\frac{w_{ha}}{w_2}\right)(h_{ha} - h_2) \tag{22}$$

and

$$p_3 = p_2\left(1 - \frac{\Delta p_{rc}}{p_2}\right) \tag{23}$$

where the regenerator cold-side pressure loss ratio Δp_{rc} is input. Then

$$t_3 = T(h_3, p_3) \tag{24}$$

Note that with no regenerator (i.e., $\epsilon = 0$ and $\Delta p_r / p_2 = 0$), the variable values at station 3 are equal to those at station 2.

The hydrogen is then used to regeneratively cool the nozzle and the combustion chamber. Fuel cooling-jacket temperature rise $\Delta t_{h,f}$ and pressure loss ratio $\Delta p_{h,f}/p_3$ are input, and the cooling-jacket exit conditions are given by

$$t_4 = t_3 + \Delta t_{h,f} \tag{25}$$

$$p_4 = p_3\left(1 - \frac{\Delta p_{h,f}}{p_3}\right) \tag{26}$$

$$h_4 = H(t_4, p_4) \tag{27}$$

$$s_4 = S(t_4, p_4) \tag{28}$$

Some of the hydrogen leaving the cooling jacket is bypassed around the turbines, and the hydrogen flow rate entering the turbine system is the same as that previously determined for the regenerator hot side. That is,

$$w_5 = w_{19} \tag{29}$$

The state conditions at station 5 are the same as those at station 4.

There are three options for specifying the turbine system configuration: (1) series with oxidizer-pump drive turbine first, as shown in figure 1, (2) series with fuel-pump drive turbine first, and (3) parallel, as shown in figure 2. For any of these configurations the turbine power requirements are

$$P_{tf} = w_2 \Delta h_{tf} \tag{30}$$

$$P_{to} = w_{12} \Delta h_{to} \tag{31}$$

and the turbine specific works are

$$\Delta h_{tf} = \frac{P_{tf}}{w_{tf}} \tag{32}$$

$$\Delta h_{to} = \frac{P_{to}}{w_{to}} \tag{33}$$

For the series arrangements

$$w_{tf} = w_{to} = w_5 \tag{34}$$

For the parallel arrangement the flow w_5 is split between the two turbines on the basis of equal pressure ratios (i.e., equal ideal specific works). This yields

$$\Delta h_{td,f} = \Delta h_{td,o} \tag{35}$$

$$\frac{P_{tf}}{w_{tf}\eta_{tf}} = \frac{P_{to}}{w_{to}\eta_{to}} \tag{36}$$

$$\frac{w_{tf}}{w_{to}} = \frac{P_{tf}\eta_{to}}{P_{to}\eta_{tf}} \tag{37}$$

and since

$$w_5 = w_{tf} + w_{to} \tag{38}$$

we get

$$w_{tf} = \frac{P_{tf}\eta_{to}}{P_{to}\eta_{tf} + P_{tf}\eta_{to}} w_5 \tag{39}$$

and

$$w_{to} = w_5 - w_{tf} \tag{40}$$

With the required specific work now determined for each of the turbines, the solution procedure is the same for each of the two turbines in any of the three arrangements. An iteration to determine turbine pressure ratio is required, since GASPLUS does not provide a means for calculating pressure with enthalpy and entropy given as the independent variables. Using the turbine arrangement in figure 1 as an example, a pressure ratio p_6/p_5 across the oxidizer turbine is assumed and

$$p_6 = p_5\left(\frac{p_6}{p_5}\right) \tag{41}$$

$$h_{6,id} = H(s_5, p_6) \tag{42}$$

$$\Delta h_{to} = \eta_{to}(h_5 - h_{6,id}) \tag{43}$$

The turbine pressure ratio is varied and equations (41) to (43) are repeated until the calculated value of turbine specific work Δh_{to} from equation (43) matches the required value from equation (33). Then,

$$h_6 = h_5 - \Delta h_{to} \tag{44}$$

$$t_6 = T(h_6, p_6) \tag{45}$$

$$s_6 = S(t_6, p_6) \tag{46}$$

The fuel-turbine inlet conditions at station 7 are the same as the oxidizer-turbine exit conditions at station 6. A similar procedure for the fuel turbine then yields p_8, h_8, and t_8. In the parallel arrangement the inlet conditions for the fuel turbine are those at station 5 rather than those at station 7.

Where the turbines are in series, the regenerator hot-side inlet conditions (station 19) are equal to the turbine-system exit conditions (station 8). With the turbines in parallel the

exit flows are mixed such that

$$p_{19} = p_6 = p_8 \tag{47}$$

$$h_{19} = \frac{w_c h_6 + w_h h_8}{w_5} \tag{48}$$

$$t_{19} = T(h_{19}, p_{19}) \tag{49}$$

The calculated value of h_{19} is compared with the estimate used for the regenerator cold-side evaluation in equation (22), and equations (22) to (48) are iterated to convergence. Regenerator hot-side exit conditions can then be determined from the following:

$$p_9 = p_{19}\left(1 - \frac{\Delta p_{rh}}{p_{19}}\right) \tag{50}$$

$$h_9 = h_{19} - \epsilon(h_{19} - h_2) \tag{51}$$

$$t_9 = T(h_9, p_9) \tag{52}$$

The regenerator hot-side exit flow is then mixed with the turbine bypass flow

$$w_{10} = w_5 + x_{bb} w_2 = w_2 \tag{53}$$

$$p_{10} = p_9 \tag{54}$$

$$h_{10} = \frac{w_5 h_9 + x_{bb} w_2 h_4}{w_{10}} \tag{55}$$

$$t_{10} = T(h_{10}, p_{10}) \tag{56}$$

The fuel-stream pressure loss ratio $\Delta p_{f,\mathrm{loss}}/p_{10}$, which is an input representing the sum of the line, valve, and injector pressure losses, is assumed to occur between the mixer and the combustion chamber. A combustion chamber injection pressure for the fuel is then determined as

$$p_{f,\mathrm{in}} = p_{10}\left(1 - \frac{\Delta p_{f,\mathrm{loss}}}{p_{10}}\right) \tag{57}$$

The fuel-pump pressure p_2 is varied and equations (16) to (57) are repeated until the calculated value of $p_{f,\mathrm{in}}$ from equation (57) matches the input value of p_{in}. This iteration is controlled by the CONTIN subroutine described in reference 6. Then

$$h_{f,\mathrm{in}} = h_{10} \tag{58}$$

and

$$t_{f,\mathrm{in}} = T(h_{f,\mathrm{in}}, p_{\mathrm{in}}) \tag{59}$$

The combustion and nozzle expansion analyses are performed by using the program CEC, described in references 3 and 4, as a subroutine. All the capabilities of the CEC program are available to this analysis (i.e., infinite-area or finite-area combustion, equilibrium or frozen expansion). The analysis methods are presented in the references and are not described herein. The CEC program uses an ideal-gas enthalpy reference temperature of 536.67 °R, which differs from the GASPLUS reference temperatures. Therefore, the CEC reactant enthalpies must be corrected. Furthermore, the enthalpy gained by the hydrogen during regenerative cooling of the nozzle and the combustion chamber must be accounted for. This is modeled herein by subtracting that enthalpy gain from the hydrogen enthalpy. Therefore, the CEC reactant enthalpies are

$$h_{f,\mathrm{CEC}} = h_{f,\mathrm{in}} - H(536.67\ ^\circ\mathrm{R},\ 14.696\ \mathrm{psia}) - (h_4 - h_3) \tag{60}$$

$$h_{o,\mathrm{CEC}} = h_{o,\mathrm{in}} - H(536.67\ ^\circ\mathrm{R},\ 14.696\ \mathrm{psia}) \tag{61}$$

It is recognized that the CEC program thermodynamic data are for normal hydrogen whereas the rocket-engine fuel is parahydrogen. Over the temperature range of combustion and expansion, the difference in the computed value of ideal specific impulse would be very small, if not insignificant. The ideal specific impulse computed by the CEC program is then multiplied by the thrust correction factor to yield the actual specific impulse.

Split Expander

The split expander cycle, shown in figure 2, differs from the full expander cycle in that the hydrogen flow is split in the fuel-pump system with part of it bypassing directly to the combustion chamber. In the low-pressure fuel pump all of the hydrogen is pumped to intermediate pressure p_{15}, where

$$p_{15} = \frac{p_{\mathrm{in}}}{\left(1 - \frac{\Delta p_{b,\mathrm{loss}}}{p_{15}}\right)} \tag{62}$$

The combustion chamber injection pressure p_{in} and the hydrogen-bypass-stream pressure loss ratio $\Delta p_{b,\mathrm{loss}}/p_{15}$, which represents the sum of the line, valve, and injector pressure losses, are both inputs. Pump exit conditions are

$$h_{15,\mathrm{id}} = H(s_1, p_{15}) \tag{63}$$

$$\Delta h_{fp,lp} = \frac{h_{15,id} - h_1}{\eta_{fp,lp}} \tag{64}$$

$$h_{15} = h_1 + \Delta h_{fp,lp} \tag{65}$$

$$t_{15} = T(h_{15}, p_{15}) \tag{66}$$

$$s_{15} = S(t_{15}, p_{15}) \tag{67}$$

The flow leaving the low-pressure fuel pump is split as specified by the input bypass fraction x_{jbp}. An amount, termed jacket bypass flow w_{15}, where

$$w_{15} = x_{jbp} w_1 \tag{68}$$

bypasses the regenerator, the nozzle/chamber cooling jacket, and the turbines and rejoins the main flow downstream in the mixer. The flow rate and enthalpy leaving the mixer are

$$w_{10} = w_5 + x_{fib} w_2 + w_{15} \tag{69}$$

and

$$h_{10} = \frac{w_5 h_9 + x_{fib} w_2 h_4 + w_{15} h_{15}}{w_{10}} \tag{70}$$

For this cycle the corrected fuel enthalpy for the CEC program is

$$h_{f,CEC} = h_{f,inj} - H(536.67\ ^\circ\text{R},\ 14.696\ \text{psia}) - (1 - x_{jbp})(h_4 - h_3) \tag{71}$$

The analysis of the high-pressure fuel pump is the same as that for the full expander cycle except that the pump inlet conditions are those for station 15 rather than station 1.

Dual Expander

The fuel-system analysis for the dual expander cycle, shown in figure 3, is the same as that for the full expander cycle except for the simplification of having only one turbine in the system. Therefore, the variable values at stations 6 to 8 and 19 are identical.

The oxidizer system is similar to the fuel system except that there is no regenerator. The oxygen-pump exit pressure p_{13} must be determined iteratively in the same manner as the hydrogen-pump exit pressure. An estimated value of p_{13} is used to determine oxidizer-pump exit conditions at station 13 in the same manner as for the full expander cycle. The oxygen is then used to regeneratively cool the nozzle. The cooling-jacket temperature rise Δt_{ocj} and the pressure loss ratio $\Delta p_{ocj}/p_{13}$ are input, and the nozzle cooling-jacket exit conditions are given by

$$t_{16} = t_{13} + \Delta t_{ocj} \tag{72}$$

$$p_{16} = p_{13}\left(1 - \frac{\Delta p_{ocj}}{p_{13}}\right) \tag{73}$$

$$h_{16} = H(t_{16}, p_{16}) \tag{74}$$

$$s_{16} = S(t_{16}, p_{16}) \tag{75}$$

Some of the oxygen leaving the cooling jacket is bypassed around the turbine, and the flow through the turbine is

$$w_{17} = w_{12}(1 - x_{otb}) \tag{76}$$

where x_{otb} is the input oxidizer-turbine bypass ratio. The oxidizer-turbine required specific work is

$$\Delta h_{ot} = \frac{P_{ot}}{w_{17}} = \frac{w_{12} \Delta h_{tp}}{w_{17}} \tag{77}$$

and a turbine pressure ratio p_{17}/p_{16} is estimated. Then,

$$p_{17} = p_{16}\left(\frac{p_{17}}{p_{16}}\right) \tag{78}$$

$$t_{17,id} = H(s_{16}, p_{17}) \tag{79}$$

$$\Delta h_{ot} = \eta_{ot}(h_{16} - h_{17,id}) \tag{80}$$

the turbine pressure ratio is varied, and equations (78) to (80) are repeated until the calculated turbine specific work Δh_{ot} from equation (80) matches the required value from equation (77). The turbine-exit enthalpy and temperature are then given by

$$h_{17} = h_{16} - \Delta h_{ot} \tag{81}$$

$$t_{17} = T(h_{17}, p_{17}) \tag{82}$$

The turbine flow and the bypass flow then mix

$$w_{18} = w_{12} \tag{83}$$

$$p_{18} = p_{17} \tag{84}$$

$$h_{18} = \frac{w_{17} h_{17} + x_{otb} w_{12}}{w_{18}} \tag{85}$$

$$t_{18} = T(h_{18}, p_{18}) \tag{86}$$

The oxidizer-stream pressure loss ratio $\Delta p_{o,loss}/p_{18}$ is assumed to occur between the mixer and the combustion chamber. A combustion chamber injection pressure for the oxygen is then

determined as

$$p_{o,inj} = p_{18}\left(1 - \frac{\Delta p_{o,loss}}{p_{18}}\right) \tag{87}$$

and compared with the input value of combustion chamber injection pressure p_{inj}. The oxidizer-pump exit pressure p_{11} is varied until the calculated and input injection pressures match. Then,

$$h_{o,inj} = h_{18} \tag{88}$$

$$t_{o,inj} = T(h_{o,inj}, p_{inj}) \tag{89}$$

and the corrected oxygen enthalpy for use in the CEC program is

$$h_{o,CEC} = h_{o,inj} - H(536.67\ ^\circ\text{R}, 14.696\ \text{psia}) - (h_{16} - h_{13}) \tag{90}$$

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15 Supplementary Notes

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16 Abstract

This analysis and this computer code apply to full, split, and dual expander cycles. Heat regeneration from the turbine exhaust to the pump exhaust is allowed. The combustion process is modeled as one of chemical equilibrium in an infinite-area or a finite-area combustor. Gas composition in the nozzle may be either equilibrium or frozen during expansion. This report, which serves as a users guide for the computer code, describes the system, the analysis methodology, and the program input and output. Sample calculations are included to show effects of key variables such as nozzle area ratio and oxidizer-to-fuel mass ratio.

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